FASTC-ID(RS)T-0787-91

FOREIGN TECHNOLOGY DIVISION

HYBRID GUIDANCE FOR MANEUVERING FLIGHT VEHICLES

by

Ling Dehai

HUMAN TRANSLATION

FASTC-ID(RS)T-0787-91 4 February 1992

HYBRID GUIDANCE FOR MANEUVERING FLIGHT VEHICLES

By: Ling Dehai

English pages: 18

Source: Yuhang Xuebao, Nr. 2, 1987, pp. 44-52

Country of origin: China
Translated by: SCITRAN
F33657-84-D-0165
Requester: FASTC/TTTAV/Capt J. Michael Phillips
Approved for public release; Distribution unlimited.

GRAPHICS DISCLAIMER

All figures, graphics, tables, equations, etc. merged into this translation were extracted from the best quality copy available.

ABSTRACT This article puts forward one type of guidance option related to the maneuver programing of reentering flight vehicles. The option in question makes use of a radiation sensing system, precisely determining the location of the starting point of the reentering flight vehicle. Use is also made of strap-on type inertial guidance systems to control the impact points of flight vehicles.

KEY TERMS Reentry Vehicle, Maneuvering Flight, Hybrid Guidance, Programmed Guidance, Strapdown Inertial Guidance, Map Matching Guidance, Radio Altimeter, Application, Digital Simulation

I. INTRODUCTION

Maneuvering flight is an important direction in the development of the technology of strategic flight vehicles. It is possible to use it in sudden defense, adding programing, raising precision, and recovery. Maneuvering flight tests are capable of giving complete checks in all such areas as flight vehicle propulsion systems, materials structure, heat resistance systems, stability, guidance circuits, and attitude control systems. It is capable of carrying out experiments on different technological levels. Options which are relatively simple and easy to carry out are to make the reentering flight vehicle, within the longitudinal plane, fly maneuvers according to preset programs. In general, it is necessary to complete the several steps below:

1. Going through precise inertial guidance or celestial-inertial guidance systems, guidance is carried out on the flight vehicle in the main power stage, with controls and corrections to the free section flight orbital plane, such that the trajectory is capable of entering into the predicted reentry corridor. As far as initial flight tests are concerned, it is possible to make use of inertial guidance systems which already exist, relaxing the requirements for launch plane controls and lateral impact point deviations.

2. Use is made of radar altimeters installed aboard the missile or other radiation systems, autonomously making precise determinations of the locations of control points. In this is included precise determinations of predicted control altitudes and descent paths at the

altitudes in question. This requires a high altitude altimeter with an operating range of from 45 Km to 150 Km, providing height measurement information in a dispersed way. Perhaps one opts for the use of other radiation systems, supplying information on measured distances corresponding to fixed markers on the ground.

3. From control start points, that is, setting out from the take off points for flight maneuvering, the strapdown inertial guidance option, which makes use of inertial measurements to compose and set up the flight maneuvering programs for reentering flight vehicles, in the initial iteration of flight tests, is capable of opting for the use of a pure strapdown inertial guidance plan, and, after the conditions become familiar, add on an imagery matching terminal guidance system.

II. DETERMINING THE SPACIAL LOCATIONS OF CONTROL START POINTS

The determination of the spacial locations of control start points is making use of the main thrust section guidance information, information on the measurement of heights and distances supplied by the radar altimeters or other radiation systems on the flight vehicle in the free section, and, during the process of the flight of the 45
flight vehicle, autonomously making a fix of the precise location of control start points. Speaking in terms of this method, it is possible to make use of information obtained from isochronous measurements of height and distances, perhaps making use of information obtained from time measurements at equal altitudes and equal distances. One takes these types of information and carries out comparisons with parameters for standard orbits, precisely determining the control start altitude that was predicted and the path of descent for the altitude in question.

1. ISOCHRONOUS MEASUREMENTS OF ALTITUDE AND DISTANCE

After winding up the main thrust section of the flight of the vehicle, one then enters into the passive section of the flight. At this time, the radar altimeters installed on flight vehicles begin to work, obtaining information on measurements of height and distance corresponding with the surface of the ground or measured to markers on the ground. If the flight vehicle flies strictly on a standard orbit, and, in conjunction with that, there are no errors in the measurements of height and distance, in that case, the data for measurements of height and distance will also be a series ot ideal values. Actually, flight vehicles will, more or less, deviate from standard orbits. Radiation systems will also have measurement noise. At the current time, going through measurement data in order to estimate the deviation of burnout point parameters for the main propulsion stage, one can carry on a step further and calculate the index deviation associated with flight vehicle reentry maneuver control start points, such as deviations in path of descent.

In the case of flight vehicle motions, one opts for the use of the model below

$$\dot{\bar{x}} = \bar{f}(\bar{x}, \bar{w}, t) \qquad \bar{w} = \bar{g}(\bar{x}, \bar{u}) \tag{1}$$

Here, $\bar{x}$ is the vector of state composed of the position vector $\bar{r}$ and the velocity vector $\bar{v}$. $\bar{w}$ is the inertial sensing output information. $\bar{u}$ is the control force or interference force vector.

On the other hand, the radiation sensing systems on the flight vehicle obtain measurement information

$$\bar{z} = \bar{h}(\bar{x}, t) + \bar{u} \tag{2}$$

In this, $\bar{z}$ is the measurment output. $\bar{u}$ is the measurement noise.

Corresponding to standard orbits, equations are set up for state interference and measurement interference

(3)

Here, $A = \left[\frac{\partial \bar{f}}{\partial \bar{x}}\right]$, $B = \left[\frac{\partial \bar{f}}{\partial \bar{w}}\right]$, $C = \left[\frac{\partial \bar{h}}{\partial \bar{x}}\right]$

which are the corresponding partial derivative matrices. $\hat{v}$ is the noise.

Assume that the instant the flight vehicle takes off is t_O, the instant of standard burn out is $t_{K(illegible)}$, the instant of actual burn out is $t_{k(illegible)}$, and the instants of separated or discrete measurements are t_i ($t_i >_K$; $i = 1,2,...$). When flight vehicles reach control start points for reentry maneuvering, the terminal index function is $L = L(\bar{x},t)$. After flight vehicles deviate from standard states, corresponding to the instant when standard control begins, t_e, there are deviations of state $\delta \bar{x}_e = \delta \bar{x}(t_e)$. This being the case, it gives rise to an index of deviation

$$\delta L = L_x \delta \bar{x}_e \qquad (4)$$

Here, $\bar{L}_x$ is the determinant vector which is formed by the partial derivatives of the components for the state $\bar{x}$ versus the function L. What must be paid attention to in $\delta \bar{x}_{e(illegible)}$ is the fact that it is given rise to by the main thrust phase interference $\delta \bar{w}$, and it will be reflected through $\delta \bar{z}$ in the free phase. In conjunction with this, it gives rise to indices of deviation associated with reentry control start points. Going through analyses of the considerable nature of the systems and their degrees of observability, it is possible to know that $\delta \bar{r}_e$ is relatively sensitive to $\delta \bar{w}$, and that $\delta \bar{z}$ is relatively sensitive to $\delta \bar{v}_e$. Because of this, it is possible to make use of the inertial measurement $\delta \bar{w}$ from the main power phase in order to calculate $\delta \bar{r}_e$ and to make use of the measurement $\delta \bar{z}$ from the free phase to estimate $\delta \bar{v}_e$. As a result of this, one obtains a general form of $\delta \bar{x}_e$. Going a step further, one calculates δL, reconsiders time difference corrections, obtaining the full deviation ΔL associated with terminal index functions.

From the theory of linear systems, on the basis of equation (3), it is not difficult to make use of $\delta \bar{w}(t)$ to solve for $\delta \bar{x}(t_e)$ for the instant t_e. One separately obtains $\delta \bar{r}(t_e)$. The reason for this is that

$$\delta \bar{x}(t_e) = \Phi(t_e, t_0)\delta \bar{x}(t_0) + \int_{t_0}^{t_e} \Phi(t_e, \tau) B(\tau) \delta \bar{w}(\tau) d\tau \qquad (5)$$

When one pays attention to $\delta \bar{x}(t_O) = 0$ and when $t > t_{k(illegible)}$, $\delta \bar{w}(t) = 0$.

Because of this, one has

$$\delta\bar{x}_e=\delta\bar{x}(t_e)=\int_{t_0}^{t_k}\Phi(t_e,\ \tau)B(\tau)\delta\bar{w}(\tau)d\tau$$

Here, Φ ($t_{e(illegible)}$, t) is the system transfer matrix. It has the corresponding or analogous components

$$\delta\bar{r}_e=\delta\bar{r}(t_e)=\int_{t_0}^{t_k}\Phi_r(t_e,\ \tau)B(\tau)\delta\bar{w}(\tau)d\tau \tag{6}$$

Here, $\Phi_r(t_e,\ t)$ is the partitioned submatrix associated with Φ $(t_e,\ t)$. 46

In another regard, making use of $\delta\bar{x}_e$, it is possible to calculate motion parameters for any arbitrary instant in the free phase

$$\delta\bar{x}(t)=\Phi(t,\ t_e)\delta\bar{x}_e+\int_{t_e}^{t}\Phi(t,\ \tau)B(\tau)\delta\bar{w}(\tau)d\tau$$

Attention should be paid to the fact that, at this time, $t > t_{k(illegible)}$. Because of this, $\delta\bar{w}(t) = 0$. Then, one has

$$\delta\bar{x}(t)=\Phi(t,\ t_e)\delta\bar{x}_e \tag{7}$$

Making use of measurements from equations in (3), corresponding to the instant t_i, one has

$$\begin{aligned}\delta z_i&=C(t_i)\delta\bar{x}(t_i)+v_i\\&=C(t_i)\Phi(t_i,\ t_e)\delta\bar{x}_e+v_i\end{aligned} \tag{8}$$

Assume that the ith iteration of covariance measurements is $R(t_i)$. Then, the minimum variance for a number of iterations of measurement is estimated to be

$$\sum[C(t_i)\Phi(t_i,\ t_e)\delta\bar{x}_e-\delta z_i]^T R^{-1}(t_i)[C(t_i)\Phi(t_i,\ t_e)\delta\bar{x}_e-\delta z_i]=\min \tag{9}$$

In order to solve, from equation (9), for the estimated values of v_e involved, it is possible to take the matrix $C(t_i)\ \Phi(t_i, t_e)$ and divide it into two partitioned matrices $C_r(t_{i(illegible)})$ and $C_v(t_{i(illegible)})$, so as to obtain

$$C(t_i)\Phi(t_i, t_e)\delta z_e = C_r(t_i)\delta r_e + C_v(t_i)\delta v_e$$

In conjunction with this, one makes $\delta \bar{z}^*_{i(illegible)} = \delta \bar{z}_i - C_r(t_i)\ \delta \bar{r}_e$. Because of this, equation (9) becomes

$$\sum_i [C_v(t_i)\delta v_e - \delta z^*_i]^T R^{-1}(t_i)[C_v(t_i)\delta v_e - \delta z^*_i] = \min$$

Note

$$C_v = \begin{pmatrix} C_v(t_1) \\ C_v(t_2) \\ \vdots \end{pmatrix}, \quad \delta z^* = \begin{pmatrix} \delta z^*_1 \\ \delta z^*_2 \\ \vdots \end{pmatrix}, \quad R = \begin{pmatrix} R(t_1) & 0 & 0 \cdots 0 \\ 0 & R(t_2) & 0 \cdots 0 \\ \cdots\cdots\cdots & \cdots\cdots\cdots & \cdots\cdots \\ 0 \cdots\cdots & & \end{pmatrix}$$

It is then possible to solve for the estimated value of $\delta \bar{v}_{e(illegible)}$, which is

$$\delta v_e = (C_v^T R^{-1} C_v)^{-1} C_v^T R^{-1} \delta z^*$$

Or, it can be expressed as

$$\delta v_e = \left(\sum_i C_v^T(t_i) R^{-1}(t_i) C_v(t_i)\right)^{-1} \sum_i C_v^T(t_i) R^{-1}(t_i) \delta z^*_i$$

Making use of equation (4), one obtains

$$\delta L = L_v \left[\sum_i C_v^T(t_i) R^{-1}(t_i) C_v(t_i)\right]^{-1} \sum_i C_v^T(t_i) R^{-1}(t_i) \delta z^*_i \qquad (10)$$

As far as the ith order interference vector $\delta \bar{w}_i$ introduced by $\delta \bar{w}$ is concerned, it expresses the i reintegration across a certain $\delta \bar{\omega}$ area. Making use of interference principle [1] or integral expansion principle [2], it is possible to carry out an integration expansion on equation (6). After that is done, one obtains $\delta \bar{z}^*_{i(\text{illegible})} = \delta \bar{z}_i - C_r(t_i)\, \delta \bar{r}_e$. Again, substituting into equation (10), it is possible to obtain an expression with the structure below

$$\delta L = \sum_i \bar{\Psi}^i \delta z_i + \sum_j \bar{\lambda}^j \delta \varpi_j \tag{11}$$

Attention should be paid to the relationships that exist between the selection of $\bar{\lambda}^j$ in equation (11) and the integration area or interval associated with $\delta \bar{w}$. If, on the basis of the standard time interval $[t_0, t_K]$, in order to calculate $\bar{\lambda}^j$, the actual interval of interference effects is $[t_0', t_K]$, at this time, corrections should be carried out on $\bar{\lambda}^j$. The simplest method is to introduce a time correction adjustment quantity. At this time, equation (11) becomes 47

$$\Delta L = \sum_i \bar{\Psi}^i \delta z_i + \sum_j \bar{\lambda}^j \delta \varpi_j + \varepsilon(\delta t) \tag{12}$$

Here, $\bar{\Psi}^i$ and $\bar{\lambda}^j$ are installation coefficients precisely specified on the basis of calculating standard orbits. $\varepsilon(\delta t)$ is a linear function of time difference. From the permissible interference conditions, one makes precise determinations going through numerical simulations or analogues.

2. MEASUREMENTS OF TIME AT EQUAL ALTITUDES

Due to a relative similarity in methods and principles, here, one only makes a simple derivation. First of all, one takes system model

(1) and changes it into corresponding equations of high degree, introducing $\dot{H} = S(t)$ and equation (1). Dividing them by each other, one obtains

$$\frac{d\bar{x}}{dH} = \bar{f}(\bar{x}, \bar{\omega}, t)/S(t)$$

After this, it is not difficult to take the equation above and write it into a corresponding perturbation equation of higher order H

$$\begin{bmatrix} \delta T' \\ \delta \bar{r}' \\ \delta \bar{v}' \end{bmatrix} = A \begin{bmatrix} \delta T \\ \delta \bar{r} \\ \delta \bar{v} \end{bmatrix} \tag{13}$$

Here, δT, $\delta \bar{r}$, and $\delta \bar{v}$ respectively express time differences, when one reaches altitudes that are the same, position differences, as well as velocity differences. A is the time change coefficient matrix which is precisely specified by the model. Corresponding to equation (13), one sets up conjugate equations

$$\begin{bmatrix} \lambda_T' \\ \bar{\lambda}_r' \\ \bar{\lambda}_v' \end{bmatrix} = -A^T \begin{bmatrix} \lambda_T \\ \bar{\lambda}_r \\ \bar{\lambda}_v \end{bmatrix} \tag{14}$$

In this, λ_T, $\bar{\lambda}_r$, $\bar{\lambda}_v$ are conjugate variables that correspond to δT, $\delta \bar{r}$, and $\delta \vec{v}$. Making use of the Bliss formula, from equations (13) and (14), one obtains

$$\lambda_T \delta T + \bar{\lambda}_r \cdot \delta \bar{r} + \bar{\lambda}_v \cdot \delta \bar{V} \Big|_{H_0}^{H} = 0 \tag{15}$$

Appropriately selecting the terminal conditions associated with equation (14)

$$\bar{\lambda}_r(H_0) = \bar{\lambda}_v(H_0) = \bar{0}, \ \lambda_T = 1$$

Substituting into equation (15), one obtains

$$\delta T = \chi_r \cdot \delta r_o + \chi_v \cdot \delta v_o \tag{16}$$

Now, one has an appropriate number of k individual measurement points, and one obtains time deviation information δT_1, δT_2, ... δT_K. Because of this, one has the relationship below

$$\begin{bmatrix} \delta T_1 \\ \delta T_2 \\ \vdots \\ \delta T_k \end{bmatrix} = \begin{bmatrix} \chi_{r1}^T & \chi_{v1}^T \\ \chi_{r2}^T & \chi_{v2}^T \\ & \cdots \\ \chi_{rk} & \chi_{vk} \end{bmatrix} \begin{bmatrix} \delta r_o \\ \delta v_o \end{bmatrix} \tag{17}$$

Note

$$B = \begin{bmatrix} \chi_{r1}^T & \chi_{v1}^T \\ \chi_{r2}^T & \chi_{v2}^T \\ & \cdots \\ \chi_{rk}^T & \chi_{vk}^T \end{bmatrix},$$

In conjunction with this, one assumes that the measured times are all equally accurate. Because of this, one has

$$\delta T = (\chi_r^T \chi_v^T) \begin{bmatrix} \delta r_o \\ \delta v_o \end{bmatrix} = (\chi_r^T \chi_v^T)(B^T B)^{-1} B^T \begin{bmatrix} \delta T_1 \\ \delta T_2 \\ \vdots \\ \delta T_K \end{bmatrix}$$

Note

$$C^T = (C_1 C_2 \cdots C_K) = (\chi_r^T \chi_v^T)(B^T B)^{-1} B^T$$ 48

Because of this, one has

$$\delta T = \sum_{i=1}^{K} C_i \delta T_i \tag{18}$$

As far as the coefficient C_i is concerned, it is precisely determined by numerical or digital simulations. Making use of this type of method, it is easy, on the basis of a certain number of measured values associated with δT_i, to accurately make precise determinations of the time to reach the predetermined altitude. Because of this, it is then also possible to autonomously determine the height.

III. STRAPDOWN GUIDANCE OF MANEUVERING FLIGHT

As far as designing a lift force flight vehicle which carries with it control wing surfaces is concerned, the wing surfaces operate at a fixed angle, correspondingly producing a fixed trim angle of attack. From this, the control forces for maneuvering flight are produced.

1. ORBITAL MODELS

With regard to the rotating globe, it is possible to set up orbital equations associated with the passive phase of flights:

$$\begin{aligned} \dot{\bar{r}} &= \bar{v} \\ \dot{\bar{v}} &= \bar{g} + \frac{\bar{L}_o - \bar{D}}{m} + \bar{a}_q + \bar{a}_c \end{aligned} \qquad (19)$$

Here, $\bar{r}$ is the position vector. $\bar{v}$ is the velocity vector relative to the earth. $\bar{g}$ is the vector for the acceleration of gravity associated with the earth. $\bar{L}_o$ and $-\bar{D}$ are, respectively, the lift and drag vectors for the flight vehicle. $\bar{a}_q$ is the centrifugal force for a unit mass. $\bar{a}_c$ is the Keshi (phonetic, possiblly Christoffel) force for a unit mass.

Assume that $\bar{u}$ displays the directional unit vector for the axis or symmetry or zero lift axis of the flight vehicle. $\bar{v}/v$ is the unit directional vector for the velocity of the flight vehicle. C_y^{α} is the derivative of the lift coefficient versus the angle of reception a. p is the air density. S is the characteristic surface area of the flight vehicle. At this time, the direction of the lift force $\bar{L}_o$ is capable of being expressed by the use of $\left(\frac{\bar{v}}{v} \times \bar{\mu}\right) \times \frac{\bar{v}}{v}$

Because of this [3]

$$L_o = C_y^\alpha \frac{1}{2}\rho v^2 S a\left(\frac{\bar{v}}{v}\times\bar{u}\right)\times\frac{\bar{v}}{v}\Big/\sin a$$
$$\approx \frac{1}{2}S\rho C_y^\alpha(\bar{v}\times\bar{u})\times\bar{v} \tag{20}$$

In the derivation, use is made of the assumption that a is a small angle, that is, sin a $\approx$ a. Making use of the vector product formula, with slight additions to equation (20), changes yield

$$L_o = \frac{1}{2}S\rho C_y^\alpha[(\bar{v}\cdot\bar{v})\bar{u}-(\bar{v}\cdot\bar{u})\bar{v}] \tag{21}$$

The magnitude of drag force $\bar{D}$, besides having the quantity 1/2 $S\rho C_{xo}v^2$, also has the induced drag force quantity η 1/2 $S\rho C_y^\alpha v^2 a^2$ which is produced at the angle of attack which is taken when in flight. In the same way, making use of a is done with the assumption of a small angle. With the approximation formula 1-cos a $a^2/2$, the induced drag quantity changes into $\eta S\rho C_y^\alpha v^2$ (1-cos a). Attention is paid to cos a = $(\bar{v}/v)\cdot u$. Going through the necessary simplifications, one obtains

$$\bar{D} = \frac{1}{2}S\rho[(\bar{v}\cdot\bar{v})^{\frac{1}{2}}(C_{xo}+2\eta C_y^\alpha)-2\eta C_y^\alpha(\bar{v}\cdot\bar{u})]\bar{v} \tag{22}$$

49

Assume that the vector radius of flight vehicles pointed toward the center of the earth is $\bar{R}$. Then,

$$\bar{a}_q = -\bar{\omega}\times(\bar{\omega}\times\bar{R}) = \omega^2\bar{R}-(\bar{\omega}\cdot\bar{R})\bar{\omega}$$
$$\bar{a}_c = -2\bar{\omega}\times\bar{v}$$

In this way, one obtains equation (19), which is suitable for free flight phases and reentry maneuvering flight phases.

If the range of flight maneuvering is relatively small, it is possible to ignore the effects of the rotation of the earth. $\bar{a}_q$ and $\bar{a}_c$ need not be considered. At this time, equation (19) changes to be

$$\dot{r} = v$$
$$\dot{v} = g + \frac{L_{*} - D}{m} \tag{23}$$

With regard to segments of reentry orbits, they are as shown below:

(1) FREE ENTRY PHASE. The altitude is approximately 70-90 km. Flight attitude control and attitude stability systems guarantee that flight vehicles take a zero angle of attack on entry. Control wings for the next phase operate to make good preparations.

(2) CONTROL START PHASE

The altitude is approximately 35-40 km. Control wings maintain maximum control forces until lateral overloads reach maximum permissible values W = y max = 30-40g.

(3) PULLING EVEN PHASE

The flight vehicle takes the maximum permissible lateral overload W y max to operate until orbital pulling level or evening up and then stops. When it is finished, it makes the acceleration instruments aboard the flight vehicle for the axial and normal directions show detection values--the relative values for the normal and axial forces--as predetermined values, in order to guarantee that the orbital angle of inclination is zero or a certain predicted angle.

(4) LEVEL FLIGHT PHASE

Making use of detected information from acceleration instruments as well as time signal structured guidance and control equations, one carries out flight maneuvering range calculations for the path of ascent, sending out dive commands or searching for characteristic land forms to carry out map matching.

(5) DIVE PHASE.

After obtaining dive commands, operating the control wings, appropriate negative angles of attack are realized in order to dive. This is done to guarantee the necessary speed of fall.

2. GUIDANCE EQUATIONS

Assuming that one has installed three snapdown acceleration instruments on board the flight vehicle, it is possible to detect the apparent accelerations along the various axses, $\dot{W}_{x1}$, $\dot{W}_{y1}$, $\dot{W}_{z1}$. In conjunction with this, one has altitude control systems to guarantee attitude stability. Going through operations of the control wings, one maintains the appropriate angle of attack in order to satisfy the required ratio of forces between normal and axial directions[4]. At this time, one has

$$-\frac{D}{m}=\dot{W}_{x_1}\cos a-\dot{W}_{y_1}\sin a$$

$$\frac{L_o}{m}=\dot{W}_{x_1}\sin a+\dot{W}_{y_1}\cos a \qquad (24)$$

On the other hand, we know that the ratio between normal and axial forces is related to angle of attack a and Mach number M. Because of this, a is capable of being expressed as a function of the K value for the ratio of forces in the normal and axial directions and $\bar{r}$ and $\bar{v}$.

$$a=a(\bar{r},\ \bar{v},\ K) \qquad (25)$$

Using $\bar{R}$ to express the overall acceleration vector given rise to by forces at work in the air, one has, as a result,

$$\bar{R}=\frac{1}{m}(\bar{L}_o-\bar{D}) \qquad (26)$$

Giving consideration to the intstallation of acceleration instruments and the small motions of the missile body in attitudes approaching nominal states, on the acceleration instruments for the z axis direction, it is possible to have an output. At this time, the relationship between $\bar{R}$ and the amounts of acceleration detected for the various axses is

$$\bar{R} = \begin{bmatrix} \cos\alpha & -\sin\alpha & 0 \\ \sin\alpha & \cos\alpha & 0 \\ 0 & 0 & 1 \end{bmatrix} \begin{bmatrix} W_{x_1} \\ W_{y_1} \\ W_{z_1} \end{bmatrix} \tag{27}$$

At this time, the interference motion equations for equation (23) are 50

$$\begin{aligned} \delta\dot{\bar{r}} &= \delta\bar{v} \\ \delta\dot{\bar{v}} &= \delta\bar{g} + \delta\bar{R} \end{aligned} \tag{28}$$

In these,

$$\begin{aligned} \delta\bar{g} &= \left[\frac{\partial\bar{g}}{\partial\bar{r}}\right]\delta\bar{r} \\ \delta\bar{R} &= \begin{bmatrix} \cos\alpha & -\sin\alpha & 0 \\ \sin\alpha & \cos\alpha & 0 \\ 0 & 0 & 1 \end{bmatrix} \begin{bmatrix} \delta W_{x_1} \\ \delta W_{y_1} \\ \delta W_{z_1} \end{bmatrix} + \begin{bmatrix} -\sin\alpha & -\cos\alpha & 0 \\ \cos\alpha & -\sin\alpha & 0 \\ 0 & 0 & 0 \end{bmatrix} \begin{bmatrix} W_{x_1} \\ W_{y_1} \\ W_{z_1} \end{bmatrix} \\ &\times \left(\left[\frac{\partial\alpha}{\partial\bar{r}}\right]\delta\bar{r} + \left[\frac{\partial\alpha}{\partial\bar{v}}\right]\delta\bar{v} \right) \end{aligned} \tag{29}$$

Substituting into equation (28), one obtains

$$\begin{bmatrix} \delta\dot{\bar{r}} \\ \delta\dot{\bar{v}} \end{bmatrix} = \begin{bmatrix} O & I \\ P & S \end{bmatrix} \begin{bmatrix} \delta\bar{r} \\ \delta\bar{v} \end{bmatrix} + \begin{bmatrix} O \\ T \end{bmatrix} \delta\bar{W} \tag{30}$$

In this,

$$P = \left[\frac{\partial\bar{g}}{\partial\bar{r}}\right] + \begin{bmatrix} -\sin\alpha & -\cos\alpha & 0 \\ \cos\alpha & -\sin\alpha & 0 \\ 0 & 0 & 0 \end{bmatrix} \begin{bmatrix} \cos\alpha & \sin\alpha & \\ -\sin\alpha & \cos\alpha & 0 \\ 0 & 0 & 1 \end{bmatrix} \bar{R} \left[\frac{\partial\alpha}{\partial\bar{r}}\right]$$

$$S = \begin{bmatrix} -\sin\alpha & -\cos\alpha & 0 \\ \cos\alpha & -\sin\alpha & 0 \\ 0 & 0 & 0 \end{bmatrix} \begin{bmatrix} \cos\alpha & \sin\alpha & 0 \\ -\sin\alpha & \cos\alpha & 0 \\ 0 & 0 & 1 \end{bmatrix} \bar{R} \left[\frac{\partial\alpha}{\partial\bar{v}}\right]$$

$$T = \begin{bmatrix} \cos\alpha & -\sin\alpha & 0 \\ \sin\alpha & \cos\alpha & 0 \\ 0 & 0 & 1 \end{bmatrix}, \quad I = \begin{bmatrix} 1 & 0 & 0 \\ 0 & 1 & 0 \\ 0 & 0 & 1 \end{bmatrix}, \quad \delta\bar{W} = \begin{bmatrix} \delta W_{x_1} \\ \delta W_{y_1} \\ \delta W_{z_1} \end{bmatrix}$$

Setting up the conjugate equation corresponding to equation (30), one makes use of the principles of interference and obtains

$$\Delta L^{*} + L_{c}^{*} = f(\delta t) + \sum_{i=1}^{\bullet} \lambda^{i} W_{i} \qquad (31)$$

Here $L_{c}^{*} = \sum_{i=1}^{\bullet} \lambda^{i} W_{i}^{*}$. $\widetilde{W}_{i}^{*}$ is the value selected for i repeated integrations or integrals of overloads on standard trajectories. ΔL^{*} is the range deviation on the course of ascent for the course flown from the control start point to the objective point relative to the course for standard altitudes. Obviously, it is made up of two sections, that is, $\Delta L^{*} = \Delta L_{1}^{*} + \Delta L_{2}^{*}$. ΔL_{1}^{*} is the vertical range deviation on the course of ascent for the point in the main powered phase at which the engines are turned off. ΔL_{2}^{*} is the descending course deviation at a predetermined control start altitude. Making use of equation (31), it is then possible to construct control equations. Attention should be paid, in equation (25), to the precise determination of K value parameters. This is the key to guaranteeing that flight vehicles complete their programs of flight maneuvering. This will be realized by attitude control circuits from the pitch channel.

3. DIGITAL SIMULATION RESULTS

(1) With Regard to Courses of Descent Associated With Exact Fixing of Predetermined Altitudes.

If one simply makes use of altitude information measured at a few dispersed points in the orbital descent phase, below 150 km and above 45 km, it is possible to estimate the deviation values of courses of descent as they correspond to standard orbital courses of descent. In general, they are 90% - 93% of the deviations of the actual orbital courses of descent relative to standard orbital courses of descent. This is to say that the relative error with this method is 7% - 10%. If one adds measurement data for an ascent phase altitude of 150 km, it is then possible to raise the precision of estimated values, that 51
is, compensation effects associated with methods which are 95% - 97%, or a relative error for methods of less than 5%.

(2) With Regard to the Precise Fixing of the Altitudes of Anticipated Control Start Points.

Calculations clearly demonstrate that, making use of information which is measured at times when altitudes are equal, in a phase of the descent which is below 150 km and above 45 km, it is possible to precisely determine the altitude of control start points. The absolute error for this method is 0.5 meter.

(3) With Regard to Digital Simulations of Guidance Equations.

Opting for the use of simplified models. Make V--velocity, θ --trajectory angle of inclination, H--height, L--range or course of ascent, a--angle of attack, W_{x1}--the integral of overload in the longitudinal axis direction on the body of the missile, W_{y1}--the integral of overload in the lateral axis direction on the body of the missile, W_{x2} and W_{y2} are respectively the integrals for W_{x1} and W_{y1}, g_0--the acceleration of gravity at the earth's surface, C_x--drag coefficient, and C_y--lift coefficient. $Q=C_x \times \frac{\rho v^2 S}{2m}$ is the drag force for a unit mass. $Y=C_y\frac{\rho v^2 S}{2m}$ is the lift force for a unit mass. Then, the orbital equations are

$$
\begin{aligned}
&\dot{v}=-Q-g\sin\theta \\
&\dot{\theta}=\frac{Y}{v}-\left(\frac{g}{v}-\frac{v}{R_0+H}\right)\cos\theta \\
&\dot{H}=v\sin\theta \\
&\dot{L}=\frac{R_0}{R_0+H}v\cos\theta \\
&\dot{W}_{x_1}=(-Q\cos a+Y\sin a)/g_0 \\
&\dot{W}_{y_1}=(Q\sin a+Y\cos a)/g_0 \\
&\dot{W}_{x_2}=W_{x_1} \\
&\dot{W}_{y_2}=W_{y_1}
\end{aligned}
\qquad (32)
$$

First of all, as far as calculating standard orbits is concerned, it includes the first and second repeated integrals associated with overloads in axial directions and normal directions after precise determinations of control start times and the start of controls.

After that, one changes reentry parameters, still taking the control start time as the zero point. Going through adjustments of the instant of dive in order to guarantee the flight vehicle dropping on the objective point, one simultaneously calculates out the first and second repeated integrals associated with overloads in axial and normal directions from the control start point to the point of the dive. After this is done, the data is formed into a table. For example, the table of data for 5 orbits from simulation calculations is:

ΔW_{x1}	ΔW_{y1}	ΔW_{x2}	ΔW_{y2}	ΔT
−403.261	257.115	−5115.340	4312.088	37.90
−421.852	262.141	−5862.946	4791.992	39.38
−429.145	265.212	−6193.303	5013.069	40.00
−435.129	268.444	−6485.285	5218.174	40.53
−451.215	274.108	−7346.145	5767.614	42.14

One solves for the parameters

$$\lambda_{x_1} = -0.01703 \qquad \lambda_{x_2} = -0.001143$$
$$\lambda_{y_1} = -0.03824 \qquad \lambda_{y_2} = 0.001049$$
$$C = 30.49427$$

One obtains the guidance equation

$$\lambda_{x_1}\Delta W_{x_1} + \lambda_{y_1}\Delta W_{y_1} + \lambda_{x_2}\Delta W_{x_2} + \lambda_{y_2}\Delta W_{y_2} + C = \Delta T \tag{33}$$

Making use of (33), one alters the reentry parameters associated with 52
the control start point. One simulates a certain orbit. Reentry speed deviation is selected as ±5 seconds. The deviation of angle of reentry is selected as $\pm 0.5^{\circ}$, obtaining a drop point deviation which is ±500 meters. With precise simulation calculations, solving for guidance equation parameters, it is predicted that it is possible to obtain even better results in the control of drop points.

REFERENCES

[1] Ling Dehai; "On an Interference Principle for Guidance Perturbations", Astronautica Sinica, No. 1, 1981.

[2] Ling Dehai; "Integral Expansion Principles", Journal of the National University of Defense Technology, No. 3, 1983.

[3] Baker, C.D., Causey, W.E. and Ingram, H.L.; "MASCOT -- A New Concept in Guidance", NASA TM X-64573.

[4] Clapp, R.T.; "A Small 'State of the Art' Maneuverable Lifting Reentry Vehicle, AIAA Paper 65 -- 492.

DISTRIBUTION LIST

DISTRIBUTION DIRECT TO RECIPIENT

ORGANIZATION	MICROFICHE
B085 DIA/RTS-2FI	1
C509 BALLOC509 BALLISTIC RES LAB	1
C510 R&T LABS/AVEADCOM	1
C513 ARRADCOM	1
C535 AVRADCOM/TSARCOM	1
C539 TRASANA	1
Q592 FSTC	4
Q619 MSIC REDSTONE	1
Q008 NTIC	1
Q043 AFMIC-IS	1
E051 HQ USAF/INET	1
E404 AEDC/DOF	1
E408 AFWL	1
E410 ASDTC/IN	1
E411 ASD/FTD/TTIA	1
E429 SD/IND	1
P005 DOE/ISA/DDI	1
P050 CIA/OCR/ADD/SD	2
1051 AFIT/LDE	1
CCV	1
P090 NSA/CDB	1
2206 FSL	1

Microfiche Nbr: FTD92C000028
FTD-ID(RS)T-0787-91